# FINAL REPORT FOR THE UNMANNED, SPACE-BASED, REUSABLE ORBITAL TRANSFER VEHICLE "DARVES"
## Volume I: Trade Analysis and Design

A design project by students in the Department of Aerospace Engineering at Auburn University, Auburn, Alabama, under the sponsorship of NASA/USRA Advanced Design Program.

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CONTRIBUTORS

Scott H. Kester ------------------ Structures

Kathleen C. Madden --------------- Avionics

Susan J. Spry -------------------- Trajectories, Aerobrake

Tracey L. Touchton --------------- Propulsion

Christina L. Weaver -------------- Structures

# ABSTRACT

This report presents the design of an unmanned, space-based, reusable Orbital Transfer Vehicle (OTV). This OTV will be utilized for the delivery and retrieval of satellites from geosynchronous earth orbit (GEO) in conjunction with a space station assumed to be in existence in low earth orbit (LEO). The trade analysis used to determine the vehicle design is presented, and from this study a vehicle definition is given.

# TABLE OF CONTENTS

## LIST OF FIGURES

# INTRODUCTION

An Orbital Transfer Vehicle (OTV) is currently under development by the National Aeronautics and Space Administration to expand the Space Transportation System (STS) in the 1990's. This spacecraft should significantly increase payload capacity beyond the Inertial Upper Stage (IUS) through the use of cryogenic propellants. Cryogenic propulsion can provide low acceleration to very large, delicate spacecraft and has a 50% higher performance than solids or space-storable liquid fuels.

The OTV would allow round-trip capability in which the OTV returns empty or delivers a payload to the space station. This reusability and high performance of the vehicle make the OTV more cost effective than expendable solid propellant vehicles.

This report outlines the mission profile of an unmanned OTV and describes the different components of an OTV design necessary to support that mission. Required trajectories and details of the aerobraking process will be included, as will the broader subjects of propulsion and structure. Avionics, which includes guidance and control, communications, and electrical power distribution, will also be discussed. And finally, a cost analysis and time line for first OTV production will be described.

## MISSION PROFILE

This OTV has been designed to meet the anticipated need of future satellite deployment and recovery, hence the name DARVES -- Delivery and Retrieval Vehicle for Earth Space. The trend in the development of satellites seems to indicate that future satellites will be heavier (1:151). Although at present the average satellite weighs only 3000-4000 lb, DARVES has been designed with the capability to carry as much as 16,000 lb (from LEO to GEO) in order to continue to meet future demands. Since this vehicle will be space-based, the cost of repetitive delivery from earth to the space station will be eliminated. DARVES will be delivered, via the Space Shuttle, to the space station, where it will be assembled and then deployed. The vehicle will be refueled and malfunctioning modules will be replaced at the space station. Therefore, DARVES will only have to be returned to earth for major repairs and/or overhauls. Some sample missions which may be performed by this vehicle are described below.

<u>MISSION 1</u>:

| | |
|---|---|
| Payload from LEO to GEO (lb) | maximum of 16000 |
| Payload from GEO to LEO (lb) | 0 |
| Propellant required (lb) | 46692 |

<u>MISSION 2</u>:

| | |
|---|---|
| Payload from LEO to GEO (lb) | 0 |
| Payload from GEO to LEO (lb) | maximum of 18000 |
| Propellant required (lb) | 47366 |

MISSION 3:

| | |
|---|---|
| Payload from LEO to GEO (lb) | 10000 |
| Payload from GEO to LEO (lb) | 7000 |
| Propellant required (lb) | 47304 |

MISSION 4:

| | |
|---|---|
| Payload from LEO to GEO (lb) | 7000 |
| Payload from GEO to LEO (lb) | 10000 |
| Propellant required (lb) | 46891 |

MISSION 5:

| | |
|---|---|
| Payload from LEO to GEO (lb) | 12000 |
| Payload from GEO to LEO (lb) | 5000 |
| Propellant required (lb) | 47578 |

MISSION 6:

| | |
|---|---|
| Payload from LEO to GEO (lb) | 5000 |
| Payload from GEO to LEO (lb) | 12000 |
| Propellant required (lb) | 46617 |

Missions 3, 4, 5, and 6 are examples of dual missions (deployment and recovery). Similar missions can be performed as long as the propellant required does not exceed the propellant available.

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OF POOR QUALITY

# TRAJECTORIES

The move from LEO to GEO and back is accomplished using a minimum energy Hohmann Transfer about the earth. The transfer involves four major velocity changes, or delta v's: three propulsive and one aerobraking.

First delta v: From low earth orbit at an altitude of 311 miles, the OTV and its payload go through a propulsive delta v at the point designated to be the perigee of the elliptical transfer orbit. This first velocity increase allows the OTV to leave circular LEO, requiring a velocity of approximately 17028 mph, and to enter an elliptical transfer orbit with a velocity of 22329 mph.

Second delta v: After approximately 318 minutes have elapsed, DARVES reaches the apogee of its transfer orbit, which is also the equatorial node, with a much-reduced velocity of 3643 mph. At this point, another propulsive delta v is required. This second change increases the velocity to 6878 mph and the OTV enters a circular orbit at GEO altitude of 22236 miles above the earth's surface. Included in the second delta v is the necessary plane inclination change from the 28.5° inclined LEO to the 0° inclined GEO. (See Fig. 1) (2:3) The higher altitude at which this change is accomplished reduces the propellant requirement because of the lower circular velocity.

Third delta v: After the OTV has delivered its payload and perhaps retrieved a second payload to bring back to LEO for repair or replacement, the return trip begins. The third propulsive delta v is simply the second delta v described above in reverse. The amount of propellant required for this velocity change may be more or less than that required by the second, depending on the mass of any return payload.

Fourth delta v: The fourth delta v is not propulsive, but rather is accomplished aerodynamically using the aerobrake. As DARVES enters the upper atmosphere of the earth heading for a previously specified perigee altitude, drag forces acting on the brake tend to slow the vehicle down enough for the OTV to re-enter LEO at the necessary circular velocity. The fourth velocity change is the same as the first delta v described only no propellant has been used to accomplish it. Again, the amount of propellant saved by using the aerobrake depends on the mass of any return payload and the mass of the theoretically "dry" OTV (a small reserve of propellant may be present).

Calculations: The delta v calculations for a sample mission were found using the equations for sample trajectory analysis. A program was developed that allows one to vary the dry OTV mass, payload to be delivered, payload to be returned, and specific impulse of the engine. The program gives the total mass of propellant required for each particular scenario. From a known mass flow rate (which could also be entered as a variable), burn times for each of the three propulsive delta v's were also calculated. The output for several proposed missions may be found in Table 1, which compares the all propulsive missions and the aerobraked missions. Note the particularly dramatic propellant savings realized by using an aerobrake in some of the return payload missions. This program was used to determine which combinations of delivered/returned payloads were possible.

## AEROBRAKE

The role of the aerobrake becomes prominent in the last stage of the mission. After DARVES returns from GEO in its elliptical transfer orbit, it enters the upper region of the earth's atmosphere, defined as that region at or below approximately 100 nautical miles, or 115 statute miles (3:152). The low densities present are enough to create significant drag forces that slow the vehicle down. An important parameter at this point is the aerobrake's ballistic coefficient,"beta". Beta is defined as $W/C_DA$, i.e. the vehicle weight divided by the drag coefficient and aerobrake area. It is the ballistic coefficient which determines the perigee altitude required for a particular aerobrake maneuver scenario; the lower the value of beta, the greater drag force created, the higher the altitude allowed for perigee, and the less aerothermal heating on the OTV.

<u>Sizing of the aerobrake</u>: A ballistic coefficient of 2.253 $lbf/ft^2$ was set for this particular design (2:4) and assumed constant. The drag coefficient was determined from Newtonian methods (4:681). A 70° half-angle conical brake was chosen because of the reduced surface heating and favorable aerodynamic stability of that geometry (2:2). Using a dry OTV weight of 7145 lbf and a return payload of 5000 lbf, the aerobrake diameter was calculated to be 62.34

ft. This diameter is large enough to prevent impingement of the wake flow on the vehicle surface.

The OTV's center of pressure was calculated using the geometry of the aerobrake and found to be 5.493 ft off the body in front of the payload, thereby insuring stability of the OTV.

Aerobrake maneuver : Multiple pass aerobrake maneuvers are a compromise between reduced heating loads due to lower drag forces and increased heating loads due to longer total mission times. Multiple passes allow the OTV perigee altitude to be higher than what would be necessary for a single pass, thereby decreasing flight velocities of subsequent passes and contributing to reduced thermal protection system (TPS) requirements (Figure 1b). An increase in total time spent within the atmosphere may add TPS requirements as well as adding to the total mission time.

Modifications were made to an existing program, written by the Mars OTV senior design group, which analyzed an aerobraking maneuver through the Martian atmosphere, in order to analyze such a maneuver through the atmosphere of the earth. Given a particular perigee altitude, density, and return mass of the OTV, the program gives the number of passes required to reach the desired apogee at LEO. Drag forces and time spent within the atmosphere are also estimated, but the theory behind such calculations is oversimplified. However, the general trends are significant. As the perigee altitude is increased, the number of passes required is increased; drag forces decrease while passage times within the atmosphere increase with succeeding passes. Even after several passes, LEO altitude may not be exactly reached, and a small delta v may be required to raise or lower the apogee altitude and another to circularize the final orbit at perigee. The fuel required for additional delta v's would be of much smaller magnitude than the delta v's required by the basic Hohmann Transfer since such "fine tuning" of the orbit is not in the same category as the major orbit changes previously described. Multiple passes may or may not be an advantage, depending on the time limits of the mission.

Thermal Protection System : A thermal protection system is required to combat incident radiative and convective heat fluxes and loads. The guidelines in choosing such a system are reusability, current knowledge of material fabrication and applicability, and proven heat protection capability for the magnitude and duration of the predicted heat fluxes and loads (5:275). The worst case heating condition occurs during the first pass of multiple pass cases, but the use of multiple passes is highly beneficial in reducing radiative flux.

The development of a suitable thermal protection system has been studied and pronounced feasible, but details are not included in this report. The reader is referred to References 2, 5, 6, and 7 for more information.

# PROPULSION

When considering the propulsion system to be used in a new vehicle, one must consider several variables: the number of engines, the engine configuration, the type of engine, and the propellants.

<u>Number of engines</u>: For the purpose of the unmanned OTV, man-rating is unnecessary; therefore, multiple engines would be advantageous only for the purpose of reliability and life cycle length (8:1-2). In order for multiple engines to increase reliability, the system must have engine-out capability, which would require the use of gimbals on all engines to align the thrust axis through the center of mass. Although the reliability is not as high, a single engine does not require the additional weight and complexity of gimbals. A single redundant-type engine (with all components duplicated except for the thrust chamber) was originally considered for this vehicle to combine the simplicity of a single engine with the increased reliability and life cycle of a multiple-engine system. After further consideration, however, it was determined that the disadvantage of the additional weight and maintenance of the redundant system would be more significant than the advantages of having such a system. For these reasons, a simple single engine was chosen for this OTV.

Engine configuration: Two feasible configurations for the single engine system were considered. The first possibility would be to mount the engine on the same end of the vehicle as the aerobrake shield, with the nozzle extension protruding through the shield. The second option is to mount the engine at the opposite end of the vehicle from the aerobrake shield. In either configuration, the engine would be mounted along the centroid axis of the vehicle so that pivoting of the nozzle is unnecessary to maintain stability. The latter configuration would require the payload to be loaded within the structure of the vehicle. The payload would then be limited in size and shape, and robotics would be required to load and unload the payload. The first configuration was chosen because it allows the vehicle to be compact, with most of the mass of the vehicle concentrated near the aerobrake shield. With this configuration, the payload will be more easily loaded and unloaded. The size and shape of the payload will be limited only by the hot wake behind the aerobrake, and the mass will be limited only by the amount of fuel available.

Type of engine: According to a NASA-sponsored study, NASA-28989 (9:1), an OTV engine should have the following characteristics:

* Expander cycle with $LO_2$ and $LH_2$
* Engine thrust of 15000 lb at a mixture ratio of 6:1
* Installed length (retracted nozzle) less than or equal to 60 in.
* Mixture ratio range of 6:1 to 7:1
* Life (cycles/hrs) greater than or equal to 300/10
* 2-position, contoured bell nozzle

Rocketdyne's RS-44 was chosen as the engine for this OTV because it meets these requirements (10:3-4) and, according to available literature, appears to be the most developed engine of its type (Fig. 2). An increased $I_{sp}$ (481 sec) and a decreased weight (~530 lb) are two highly desirable consequences of the RS-44 being an expander cycle engine.

The extendable nozzle also helps to increase the $I_{sp}$ by allowing a higher combustion chamber pressure. Unfortunately, there are complications that arise from using a retractable/extendable nozzle. During the aerobraking maneuver, the extended portion of the nozzle must be retracted, and the opening in the aerobrake shield must be covered. After the aerobrake maneuver, the nozzle will be extended again. A mechanism will be required that can extend the nozzle, lock it into place, and retract it as necessary. The available sources of information on the RS-44 and other engines of this type offer no explanation of this mechanism or how it will operate. Therefore, it is assumed that this problem is still being studied and that a satisfactory solution has not yet been reached. For the purposes of this design project, an estimated weight of 100 lb has been added to the vehicle to account for the presence of the mechanism. The mechanism for covering the opening in the aerobrake shield will be discussed in the Structures Section.

Other specifications of the RS-44 are as follows

(10:3-5):

* Chamber pressure of 1540 psia
* Nozzle area ratio (retracted) of 225:1
* Nozzle area ratio (extended) of 625:1
* Length (nozzle retracted) of 60.00 in.
* Length (nozzle extended) of 117.03 in.
* Tank head idle of 61 lb at a mixture ratio of 2.9:1
* Pumped idle of 1500 lb at a mixture ratio of 4:1

Propellants: As mentioned above, the RS-44 is a liquid oxygen/liquid hydrogen rocket engine. The liquid oxygen tank will be elliptical with a volume of 572.1 $ft^3$ and hold 41,038.2 lbm of oxygen. The hydrogen tank will be spherical with a volume of 1552.6 $ft^3$ and hold 6862.3 lbm of hydrogen. These tanks will have to be kept at constant pressures during engine burn to ensure steady fuel flow to the engine. This pressurization will be accomplished using nitrogen as a pressurizing gas. The nitrogen tank will be spherical, with a volume of 56.05 $ft^3$ and will initially be highly pressurized to 2205 psia. The nitrogen will flow through separate regulators into the oxygen tank and the hydrogen tank. These regulators will maintain constant pressures in the propellant tanks even though the pressure in the nitrogen tank will be decreasing.

Since DARVES is space-based, storage and transfer of propellants will take place at the space station or possibly at a refueling station in the same orbit as the space station. Further study must be done, however, to develop the technology needed for the safe storage and transfer of cryogenic propellants in space. (11:105)

# AVIONICS MODULE

The avionics module for DARVES includes all guidance, navigation, and control, as well as communications and power systems necessary to ensure mission accomplishment. This module will be analyzed according to the OTV's mission capabilities and requirements. The avionics module is 14 ft in diameter by 4 ft in length and weighs approximately 1000 lbs. It is comprised of three "black boxes" which are arranged in the module as shown in Figure 3. The systems discussed herein are designed to support a total mission time of approximately two days.

Guidance, navigation, and control (GNC): This portion of the avionics system contains computers programmed for attitude control and navigation. The OTV must successfully navigate to satellite, GEO, and back to the space station without incident.

The attitude control system of a spacecraft maintains the spacecraft within allowable safety limits. The block diagram of an attitude control system is shown in Figure 4 (1:162). The basic elements of this system are sensors, actuators, control laws, and disturbance torques. Spacecraft attitude dynamics predicts the motion of the spacecraft as a result of disturbance and control torques are provided by actuators, which in this case are thrusters.

This OTV attitude control system can be classified as a three-axis stabilization system, in which the entire spacecraft is orientated toward the earth. The control torques along the three axes (yaw, pitch, roll) (1:161) are provided by a reaction control system (RCS). The RCS will render translational and rotational velocity changes. This system consists of three modules of vernier thrusters, all aft of the payload module and each having its own monomethyl hydrazine fuel and nitrogen tetroxide storage system. The vernier engine chosen for this mission is the R-1E. Each module houses four vernier thrusters, capable of providing 25 pounds of thrust. Thus the RCS can provide a total of 100 pounds of thrust for each axes' control torques. These torques are applied through the thrusters which fire based on data from sensors. The thrusters cause the OTV to change its attitude when the sensors indicate the error rate in yaw pitch or roll exceeds acceptable levels. One thruster module is mounted on each axis. This RCS, modeled after the Space Shuttle's, was chosen because it proved to be more advantageous than a momentum wheel system for this OTV design when taking size, weight and reliability into consideration (11a:20).

Along with control systems, which guide to a pre-programmed course, the GNC system must also control the burn rate, or initiate the burn when the OTV approaches apogee and perigee of elliptical transfer orbits. Furthermore, this system will have the capability of initiating controls to keep drag constant during aerobrake maneuver when the OTV encounters "density pockets" in the atmosphere. Attitude control is actually achieved by having the drag brake revolve around the hinge point. Displacement in the position of the vehicle causes a shift in the c.p. location, producing a small motion about the yaw axis and resulting in an angle of attack. This provides a small lifting capability to compensate for the unpredictable composition of the atmosphere structure (2:2). The GNC will sense force variations on the aerobrake and activate the drag brake rotation to compensate for these changes.

As a measure of safety, the space station will have the capability to override the control computer in case of emergency. Additional capabilities of the GNC will include environmental control which will be used to monitor the temperature and operating parameters of the avionics section itself (12:216), and health monitoring which will be used to oversee the conditions of all OTV subsystems.

When designing the GNC it is important to ensure the subsystem is compatible with, and can be linked to, the communication systems.

<u>Communication and data handling system (CDH)</u>: The CDH system will provide command and data links between OTV and space station as well as relay data position, etc., to the space station (1:175). This system will contain computers, recorders, measuring equipment, antenna, and a data bus

which ties the entire communication subsystem together. The communication system for DARVES is similar to the one used on Mariner 4 except there will be no need for varying the telemetry bit rates (13:78). The communication subsystem will consist of a PCM/PM phase locked telemetry and commands. It will operate on a transmitting frequency of 2300 Mc and receiving frequency of 2300 Mc (12:260). A wire antenna (13:78), 3 feet in diameter, will be mounted on the forward end of the vehicle fuselage to receive the signal, and transponders will amplify and translate the frequency of the signal. In addition to transmitting engineering and scientific data back to the space station, the communication subsystem must always carry commands from the mission director to the OTV (12:80). The mission director can remotely observe docking and/or delivery activities via the use of a video camera. The camera is located on the forward end of the fuselage near the payload docking module. All communication systems must be compatible with the tracking and guidance subsystem.

Electric power and distribution: The OTV, of course, needs power for operation of its subsystems (GNC, CDH, computers). The power subsystem must also meet power requirements to provide electric spark to the engine, extend/retract nozzle, and operate cameras and docking mechanism during payload attachment and detachment. The power requirements for the OTV will be met using a fuel cell power plant (primary) and a battery as a backup system. The fuel cell power plant was chosen as a primary system over the battery because the battery system, although tried and true, has limited capability, especially with the projected growth in power levels and life expectancy for orbiting systems (14:157). The fuel cell was chosen over a solar array because solar panels are impractical. During the aerobraking process a panel storage area would be needed. The fuel cell power plant will use a lightweight $H_2/O_2$ fuel cell shown in Figure 5. Fuel cells are advantageous for this OTV design because they have long life, high specific power, reliability, maintainability, and relatively low cost (14:159).

The fuel cell power plant is based on Bacon/hydrogen-oxygen technology but incorporates advances in material design (15:42-6). This power plant has the same basic design as the Shuttle power plant, except the OTV power plant utilizes a new lightweight alkaline fuel cell which weighs 4lb/kw versus 8lb/kw for the Shuttle fuel cell (see Figure 6) (14:164). The power plant contains (14:170):

* two stacks of 32 cells in series
* Au/Pt catalyzed electrodes
* 0.5 mm (0.020") reconstituted asbestos matrix
* plastic plated separator plates
* fiberglass epoxy frame material

The operating conditions of the power plant are:

* 355 K (180°F) nominal
* 4 atmospheres of pressure
* 27.5-32.5 volts
* 7 kW nominal power rating

The proposed fuel cell power section is capable of supplying 12 kW at peak, which corresponds to 27.5 V and 436

A, and 7 kW average power; at 2 kW the power plant provides 32.5 V and 61.5 A (15:42-6). The power plant will be housed in much the same manner as the Shuttle power plant, but due to the lightweight fuel cells being used, will weigh only 57.16 kg (126 lb) versus 91 kg for the Shuttle power plant, while providing the same amount of average power (Vol. II, p.23). Hydrogen and oxygen, stored cryogenically, are delivered to each fuel cell. On a two day mission, the power plant is estimated to use about 300 kg of $H_2$ and $O_2$, with about 400 L of water being produced. This power plant must be serviced between missions and reused until it has accumulated 2500 hours on line service (15:42-6). In the event that this system somehow fails on mission, the avionics module will contain a battery back-up system. This battery will be a small secondary Ni/Cd battery and will have a long cycle and calendar life.

The entire avionics system module will be insulated for thermal protection of the module.

# STRUCTURE

The initial configuration considered in the original design proposal was of modular construction consisting of the following modules: payload, electronics and control, propellant tanks, propulsion, and aerobrake. Modules were connected through nodes to be removed and replaced as needed. The payload was located at the front of the OTV and was connected to the electronics and control module. The electronics and control module was then connected to the propellant tank module, which consisted of two torispherical propellant tanks connected by a truss structure. This truss was then connected to the engine module. And finally, the engine module connected to the aerobrake. Much of the original structure was to be constructed of composite materials (Figure 7).

Further consideration has been given to the design of the spacecraft and several disadvantages have been found. Due to possible flow impingement during aerobraking, the position of the electronics and control module at the front of the structure was not practical. The torispherical propellant tanks did not allow for sufficient propellant capacity. And finally, the use of composite materials has been ruled out due to studies which have shown degradation of composites upon exposure to the harsh space environment and because of their lowered performance at cryogenic temperatures.

Several alternatives for the OTV configuration have been considered. The first configuration consisted of a single large spherical oxidizer tank and six smaller fuel tanks located symmetrically about the oxidizer tank. The tanks were to be connected through a composite truss structure (Figure 8). Benefits of this configuration were the movement of the vehicle's center of gravity closer to the aerobrake and the reduction of flow impingement on vehicle components. The complexity of this design and the large dry weight of the vehicle necessitated further study and a new vehicle design.

The next design considered consisted of spherical fuel and oxidizer tanks connected to each other through a truss structure. This design utilized the propellant tanks as load carrying members; unfortunately, these loads increased the tank weights severely. Also, the center of gravity location was not favorable. A semi-monocoque structure consisting of two cylindrical sections of differing diameters was decided upon to house the two propellant tanks (Figure 9). But this design did not make optimal use of the volume available.

The configuration which was finally chosen consisted of a spherical fuel tank and an elliptical oxidizer tank housed in a cylindrical semi-monocoque structure of constant radius. The components of the vehicle will now be discussed separately.

Payload/docking: The payload and docking module is designed for the secure attachment and detachment of payloads through the use of remotely-operated locking mechanisms. Detachment of the payload would be accomplished through the use of a spring loaded device to safely separate the OTV from the payload. The design load is approximately 20,000 lbs, calculated from the maximum acceleration of 4 g's that a return payload of 5,000 lbs would experience during the aerobraking manuever. The module consists of a center locking section which carries the majority of the payload forces. The payload is stabilized by four braces located on the periphery of the docking module. The module is constructed of aluminum 7075 T6. The approximate mass of the module is 200 lbm. The payloads to be transferred by the OTV would have a standard docking interface to assure proper attachment to the OTV and should be compatible with space shuttle payload attachments.

Propellant tanks: The spherical fuel tank and elliptical oxidizer tank are of approximately the same radius (Figure 10). The ratio of the oxidizer tank's semi-major axis to semi-minor axis is 2:1 -- this ratio was chosen because it requires the lowest weight for an elliptical fuel tank. This spherical/elliptical tank combination moves the center of gravity for the vehicle forward, makes optimal use of the available volume, and lowers the overall weight of the OTV structure. Since the radii of the two tanks are approximately equal, there is no need for the housing structure to "neck down" as in the previous design (Figure 9). The complexity of the housing is thereby reduced, lowering its weight and making its construction easier.

In designing the propellant tanks, the stress present in the tank skin is the primary consideration. The forces on the tank walls consist of the inertial force of the fuel during the vehicle's acceleration and the internal pressure of the propellants. The propellant tank pressures for $LH_2$ (liquid hydrogen) and LOX (liquid oxygen) are 38 psi and 42 psi respectively (16:9). Inertial forces were calculated through a numerical integration process in which the instantaneous acceleration of the vehicle was calculated at 10 sec intervals for the entire engine burn time. Then the maximum pressure was used to obtain the wall thickness of the tanks.

The $LH_2$ tank has a 7.185 ft radius and a volume of 1,552.6 $ft^3$ and is constructed of Aluminum 2219 T-87, an aluminum-copper alloy. This alloy has outstanding material properties at cryogenic temperatures and is used in the Space Shuttle external tank. Some of the properties of this alloy are listed below (17:298):

* density = 0.1 $lbm/ft^3$
* tensile strength = 85,000 psi (at -320°F)
* yield strength = 67,000psi (-320°F); 75,000psi (-423°F)
* welded strength after heat treatment is nearly 100% of parent material

The temperature of the stored $LH_2$ is -423.7°F. The tank is designed for a maximum pressure of 38 psi -- 34 psi is due to internal propellant pressure and 4 psi is caused by inertial force. A design factor of safety of 1.5 was used in calculating wall thickness. The thickness was determined to be 0.0367 in, using a conservative yield strength of 67,000 psi (17:298). The mass of the $LH_2$ tank was calculated to be 342.84 lbm based on the given radius and wall thickness.

The $LO_x$ tank is composed of the same aluminum alloy as the $LH_2$ tank. The tank has a semi-major axis of 77.86 in and a semi-minor axis of 38.90 in. The eccentricity is 0.866, and tank volume is 572.1 $ft^3$. The design pressure was 48.5 psi, with 28 psi derived from the internal pressure and the remainder from the inertial force. Tank thickness was determined to be 0.0845 in, based on a maximum stress occurring at the tank bottom. Total tank mass was 444.21 lbm based on given dimensions and wall thickness and using the surface area equation for an "oblate spheroid" (an ellipse rotated about its semi-minor axis). Temperature of the LOX is -297°F.

Both the propellant tanks must be insulated to avoid excessive thermal conductivity. A total thickness of 0.75 inches of multi-layer, double-aluminized Mylar is used for this purpose. A weight of approximately 140 lbs was determined for the insulation. Estimated boil-off rates of 0.7% per week of storage made the calculation of boil-off unnecessary due to the short duration of OTV missions (18:498).

The pressurization tank containing the pressurizing gas (nitrogen) was chosen to be spherical in shape to require the least amount of weight. The radius of this tank is 2.374 ft and volume of 56.1 $ft^3$. The tank has a design pressure of 2200 psi. Because the internal pressure is so high, and the operating temperature is relatively high, a tank of composite material was chosen. Made of kevlar with an epoxy resin, wound around a seamless aluminum tank, the composite tank has a mass of approximately 360 lbm, about half the mass of a pure aluminum tank.

Avionics Module: This module is 7 ft in radius and 4 ft in length. The module is cylindrical in shape and composed of Aluminum 7075 T6. It houses the avionics equipment and is connected to the fuselage at the forward end and the engine module at the rear.

Engine module: The engine module contains the RS-44 engine and all its auxiliary equipment. This module is cylindrical in shape and designed to separate from the rest of the OTV for engine servicing and/or replacement. An approximate radius is 4 ft and a length of 5 ft. The engine module is also the point of attachment for the aerobrake; consequently, the module must transmit all of the major forces that will act on the OTV to the rest of the structure. The module is composed of Aluminum 7075 T6, which has the high yield strength needed. A mass estimate was not made for this module separately but was included as part of the overall mass estimate of the structure.

Vehicle fuselage: The propellant tanks and the pressurization tank are encased in a protective structure, which attaches to the payload docking mechanism at the forward end and the avionics module at the aft end (see Figures 10, 11, 12). This housing, or "fuselage", is a semi-monocoque structure composed of Aluminum 7075 T6, which has the following physical characteristics:

* density = .1 $lbm/in^3$
* tensile strength of 83,000 psi (at room temp.)
* yield strength of 73,000 psi (at room temp.)

The fuselage has a constant radius of 7.4 ft and a length of 21 ft. This fuselage consists of a thin, cylindrically-shaped stressed-skin supported by longitudinal stiffeners and radial rings. The stiffeners increase the buckling strength of the vessel and act as thin columns with a length equal to the spacing of the rings. The rings and stiffeners therefore effectively break up the skin into small, curved panels.

The main loads transmitted to the fuselage are the compressive forces which occur during launch and aerobraking maneuvers and the possible bending moments which may occur during the aerobrake procedure. The inertial loads from the tanks are transmitted to the fuselage through the support rings.

To minimize the thermal conductivity of the connections between the fuselage and propellant tanks, composite materials are employed. Basket-shaped, fiberglass tank supports connect to the fuselage radial rings by means of boron-aluminum metal-matrix composite tubular struts. A mass reduction of 44% is anticipated by using the composite struts rather than pure aluminum struts (19:64).

A detailed stress analysis of the fuselage and internal structure was beyond the scope of this project and is left for future study. An estimated mass for the fuselage of between 1800 and 1900 lbm is thought to be conservative.

Aerobrake: The aerobrake of the OTV is a 70 deg conical lifting brake of radius 59.1 ft. It consists of five major sections: the surface fabric, the ribbed beams supporting this fabric, the insulation between the fabric and support beams, the columns supporting the ribbed beams, and the structure supporting the entire apparatus. The surface fabric is a reflective fabric composed of silica and Nextal which are interwoven into a fiber cloth having a thickness of approximately 0.0098 in.

Eighteen sets of graphite polyimide ribbed beams act as supports for the fabric. Graphite polyimide was chosen as a material due to its low mass and ability to maintain its properties at high temperatures of up to 620°F.

To insulate the ribbed beam from potentially high skin temperatures, an insulating layer of ceramic tiles, comparable to those used on the Space Shuttle's exterior, is employed on the ribbed beams.

The ribbed beams are supported by columns, also constructed of graphite polyimide, which are then connected to the central supporting structure of the aerobrake

(20:290).

The central structure connects to the OTV on a mounting hinge located on the propulsion module. This hinge mount allows the aerobrake to be rotated with respect to the OTV longitudinal axis for aerodynamic control during the aerobraking process.

The aerobrake has a 68 in diameter aperture at its vertex which allows for operation of the extendable rocket nozzle. This opening must be covered during the aerobraking process and is a very critical area of the aerobrake since the aerobrake stagnation point is centered at this area. One design to be used for this purpose is a carbon-carbon composite shield which will be remotely secured in place prior to the aerobraking procedure. Carbon-carbon composite is utilized because of its extremely high heat resistance.

The entire aerobrake structure is estimated to weigh 2300 lbs. This figure came from estimates taken from several technical papers on aerobrake structure. No reusable aerobrake mechanism has every been flown; therefore, all design parameters are based on experimental data. Further studies including flight test are required before an operational design can be finalized.

## COST ESTIMATE

The estimations of the research and development costs and per unit costs for DARVES are given below based on a fleet size of ten units. The per unit cost includes the cost required for the Shuttle to carry the payload and fuel to the space station for the OTV to then retrieve (21:4):

Cost Estimate for OTV:
(in millions of 1986 dollars)

| | | |
|---|---|---|
| Shuttle flight to transfer fuel and payload | | 100 |
| Total Cost Estimate for Research and Development | | 655 |
| Total Cost Estimate Per Unit | | 58 |
| Aerobrake: | Research and Development | 150 |
| | Per Unit | 10 |
| Structure: | Research and Development | 40 |
| | Per Unit | 12 |
| Propulsion: | Research and Development | 200 |
| | Per Unit | 15 |
| Propellant Storage: | Research and Development | 15 |
| | Per Unit | 1 |
| Avionics: | Research and Development | 50 |
| | Per Unit | 15 |
| Final Assembly | | 5 |
| Final Testing | | 200 |

## TIME ESTIMATE

The time required for final research, construction, and testing of the OTV has been estimated and is given in Table 2. This estimate is based on the first OTV becoming operational in mid-1995. With research and development beginning in early 1988 and testing beginning in 1993, a projected time line was constructed to have the first operational OTV completed by the target date.

## SUMMARY

The Orbital Transfer Vehicle marks the beginning of the extension of man's capability past low earth orbit to geosynchronous earth orbit and beyond. The OTV allows for the benefits of the Space Shuttle and its derivatives to be fully exploited and for development of the American space operations potential to its fullest. With the direct benefits of the OTV also comes advances in the technology of aerobrake design, cryogenic fuel storage, and of other areas. The design and development of the OTV is currently underway by NASA and should be a principal focus of study in the decade to come.

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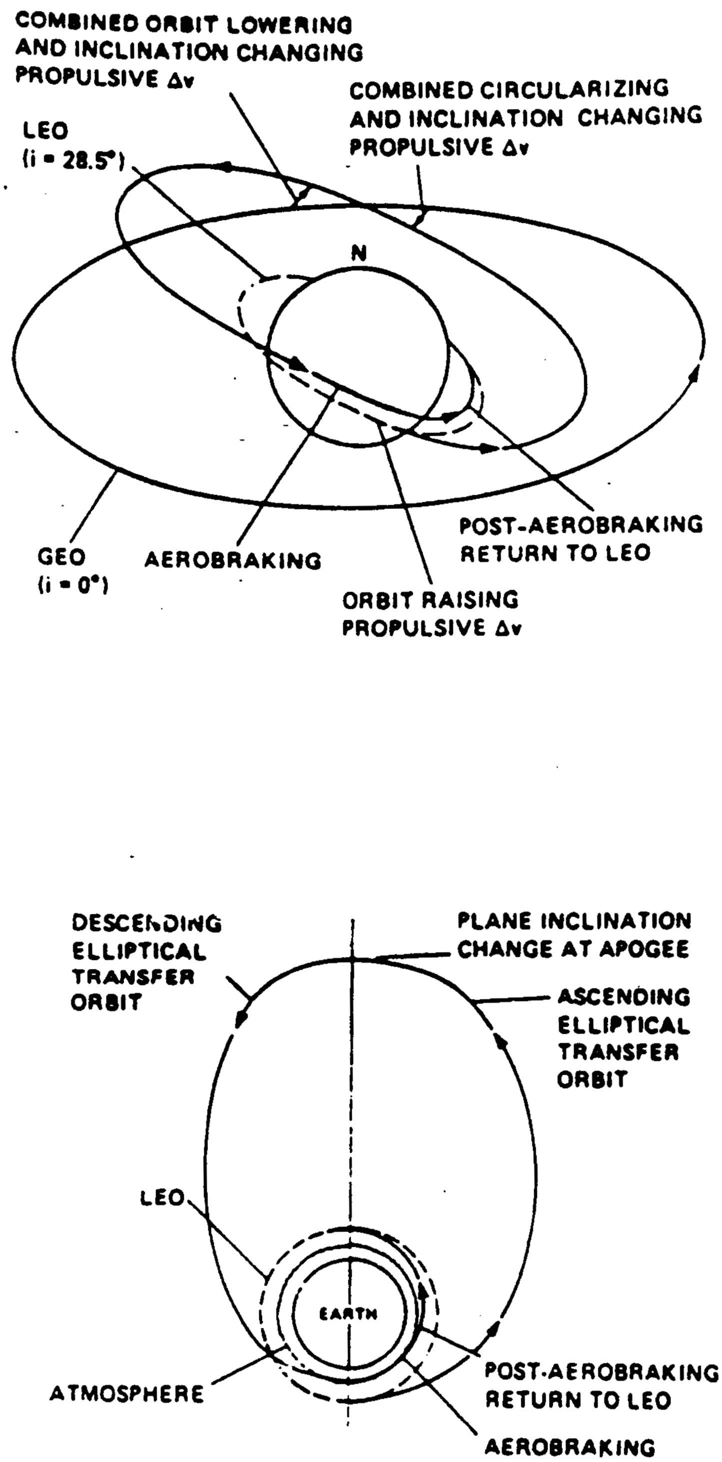


FIGURE 1. Details of the Orbital Change and Aerobraking Maneuver

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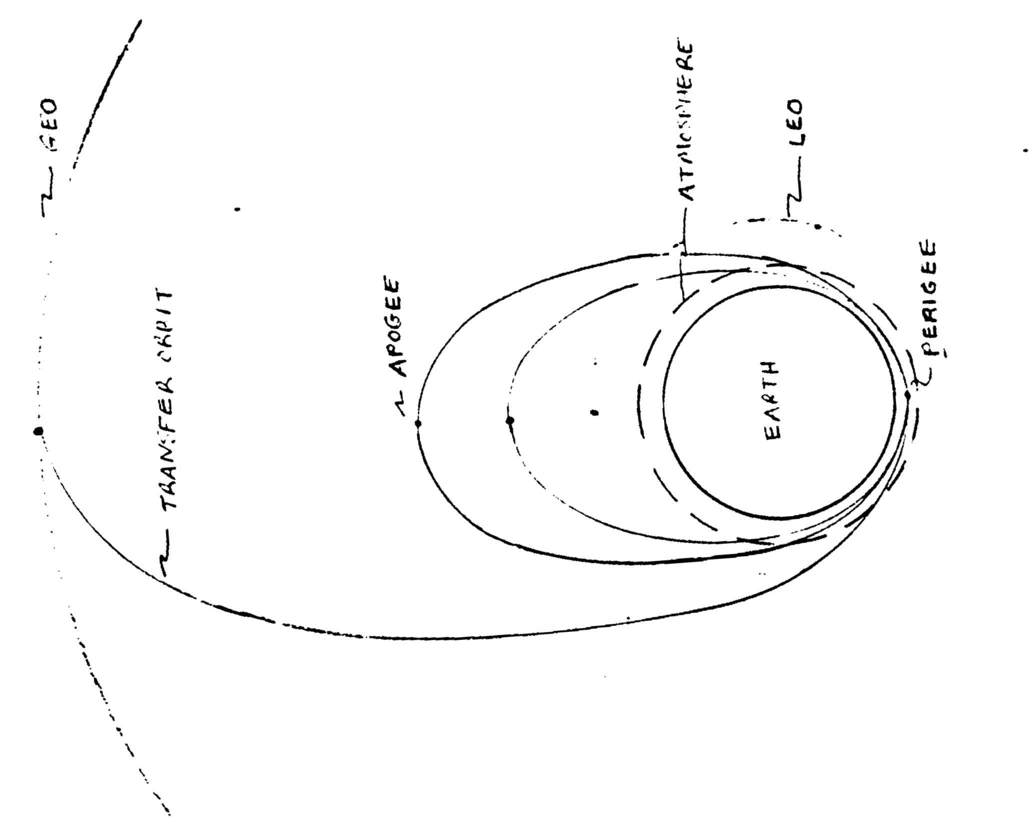
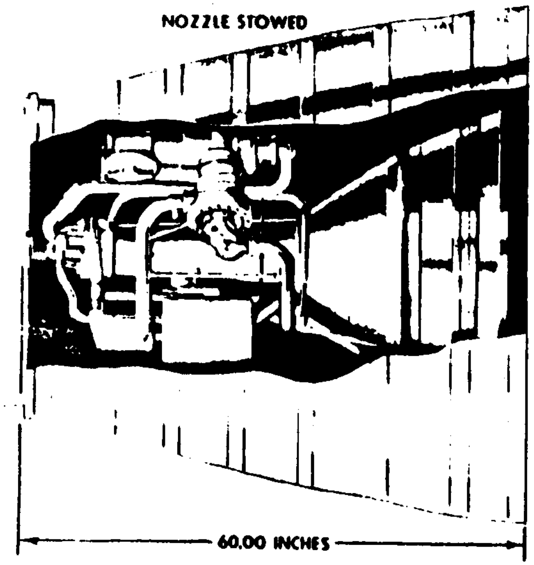



FIGURE 1b Multiple Pass Aerobrake Maneuver




# INCREASED CAPABILITY RS-44 ENGINE

NOMINAL CHARACTERISTICS

- FULL THRUST (VAC), LB ........ 15,000
- PUMPED IDLE THRUST (VAC), LB ... 1500
- MIXTURE RATIO ........ 6±1
- CHAMBER PRESSURE, PSIA ........ 1540
- EXPANSION AREA RATIO ........ 625
- SERVICE LIFE, HOURS ........ 10
- SPECIFIC IMPULSE, SEC ........ 481

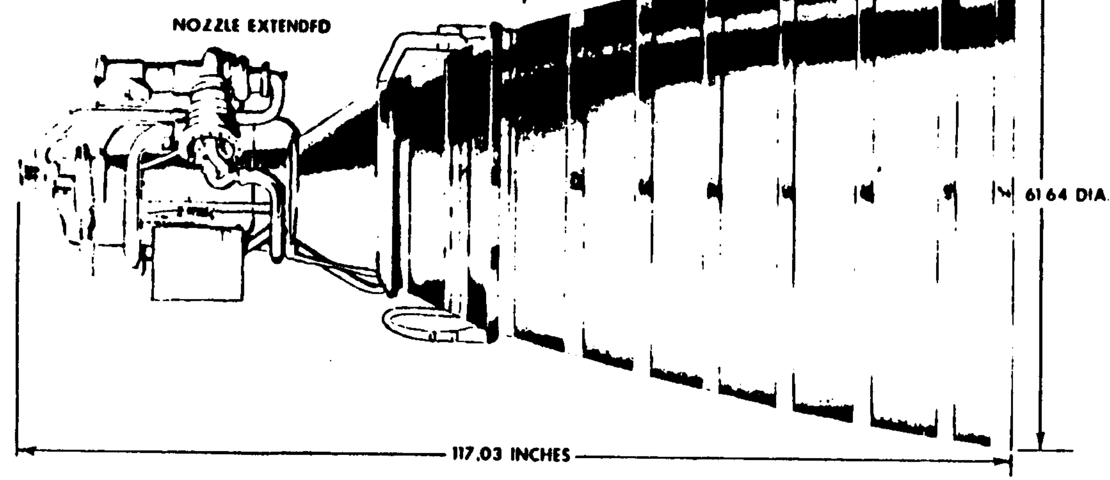



FIGURE 2. Diagram of the RS-44 Advanced Engine

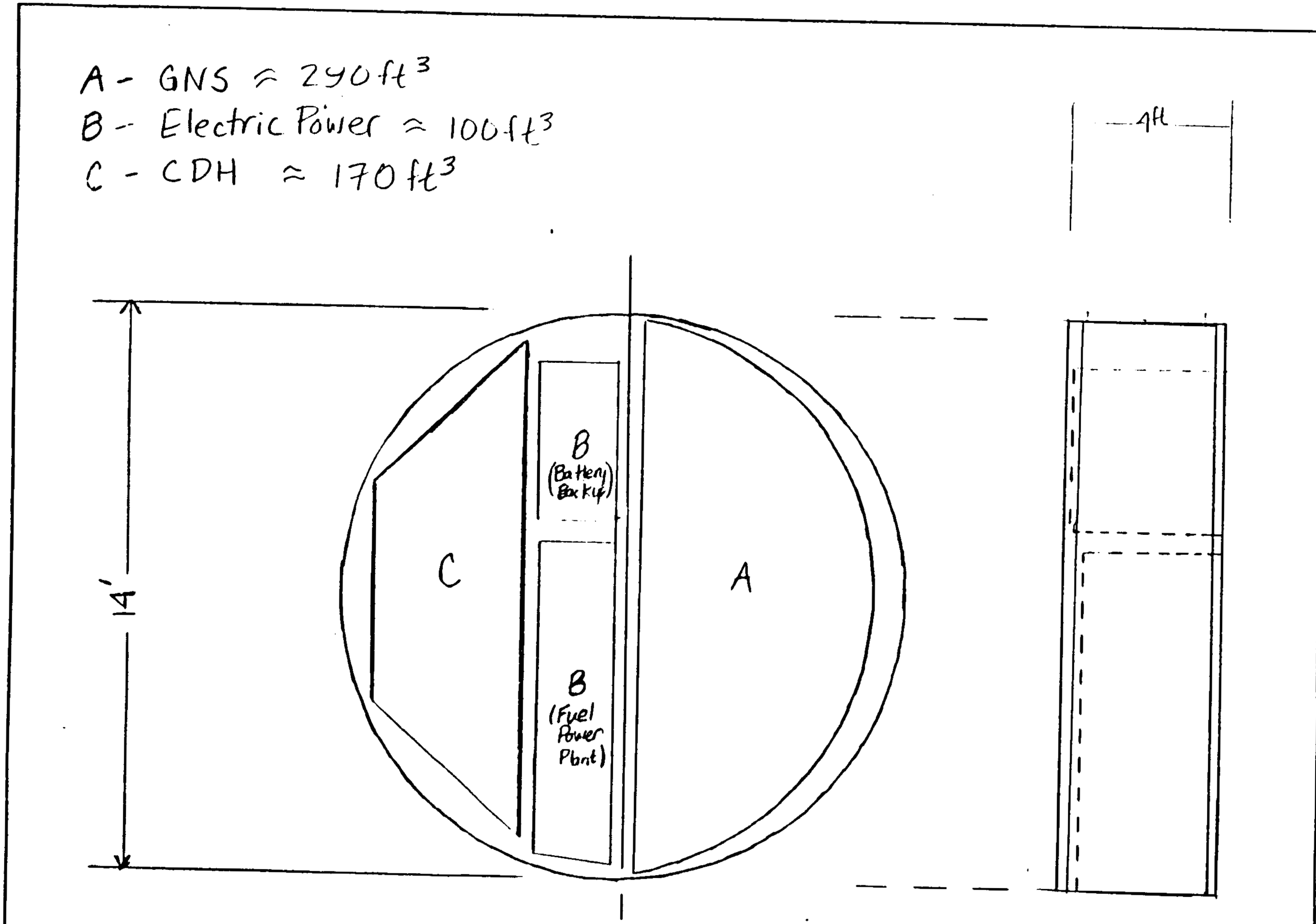



FIGURE 3. Layout of Internal Avionics Module Components

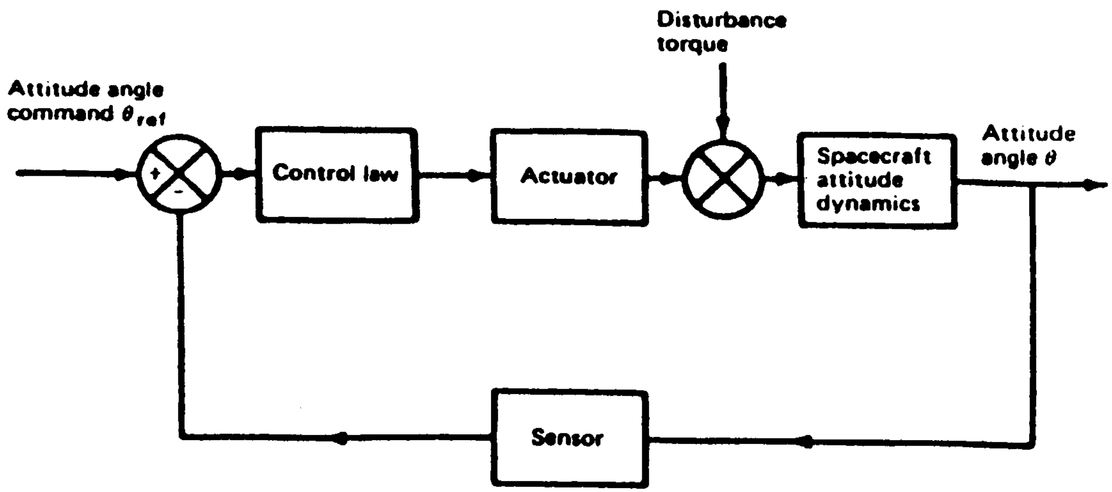



FIGURE 4. Block Diagram of Attitude Control Systems

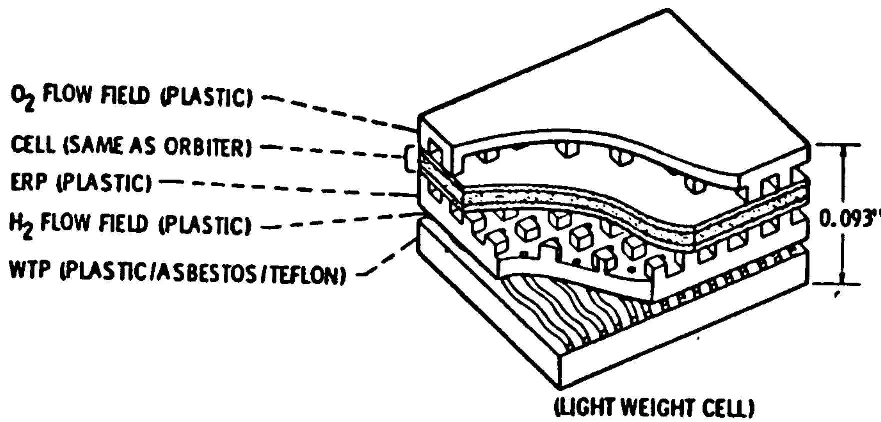



FIGURE 5. Cutaway Diagram of Lightweight $H_2O_2$ Fuel Cell

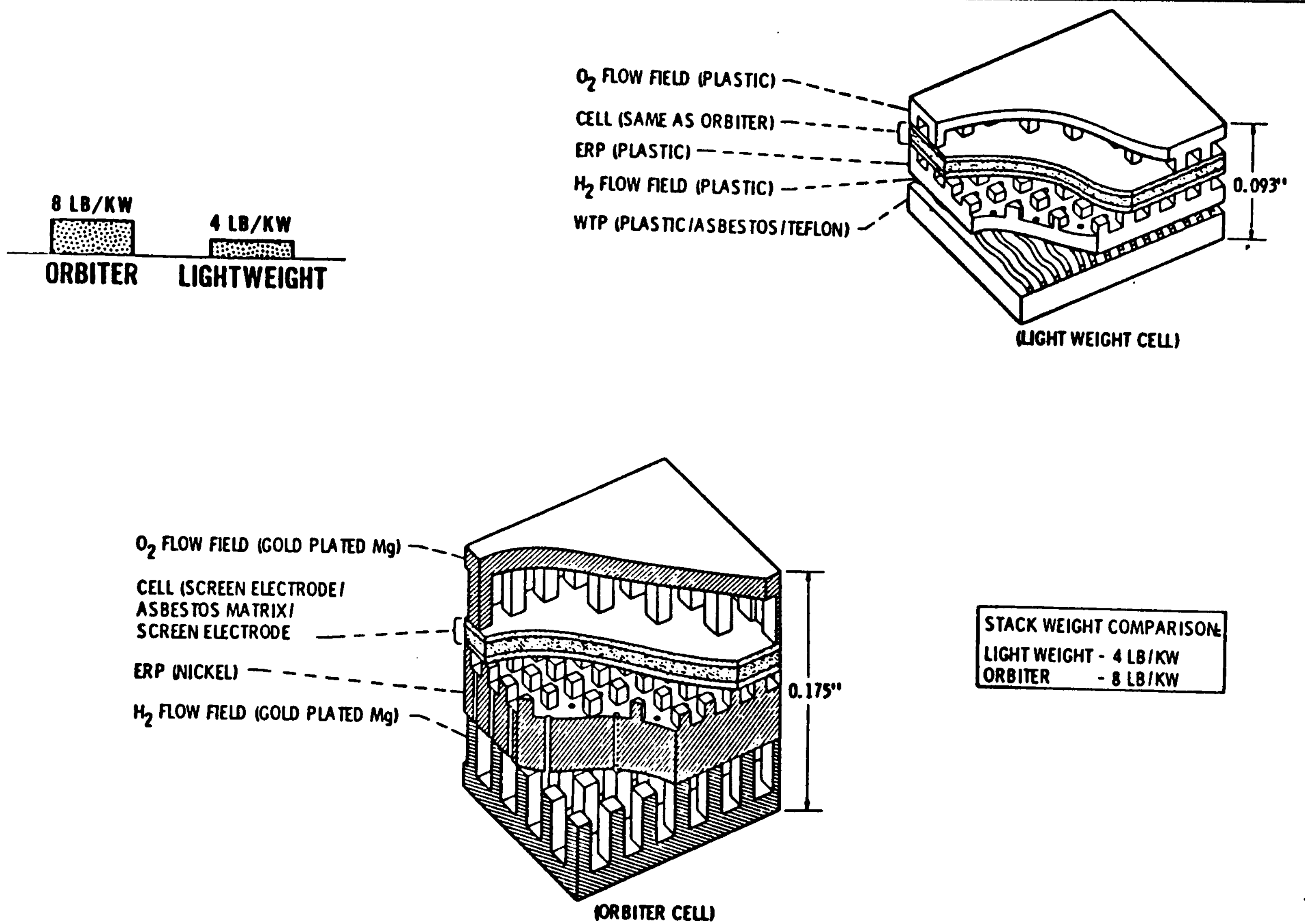



FIGURE 6. Space Shuttle/OTV Fuel Cell Comparison

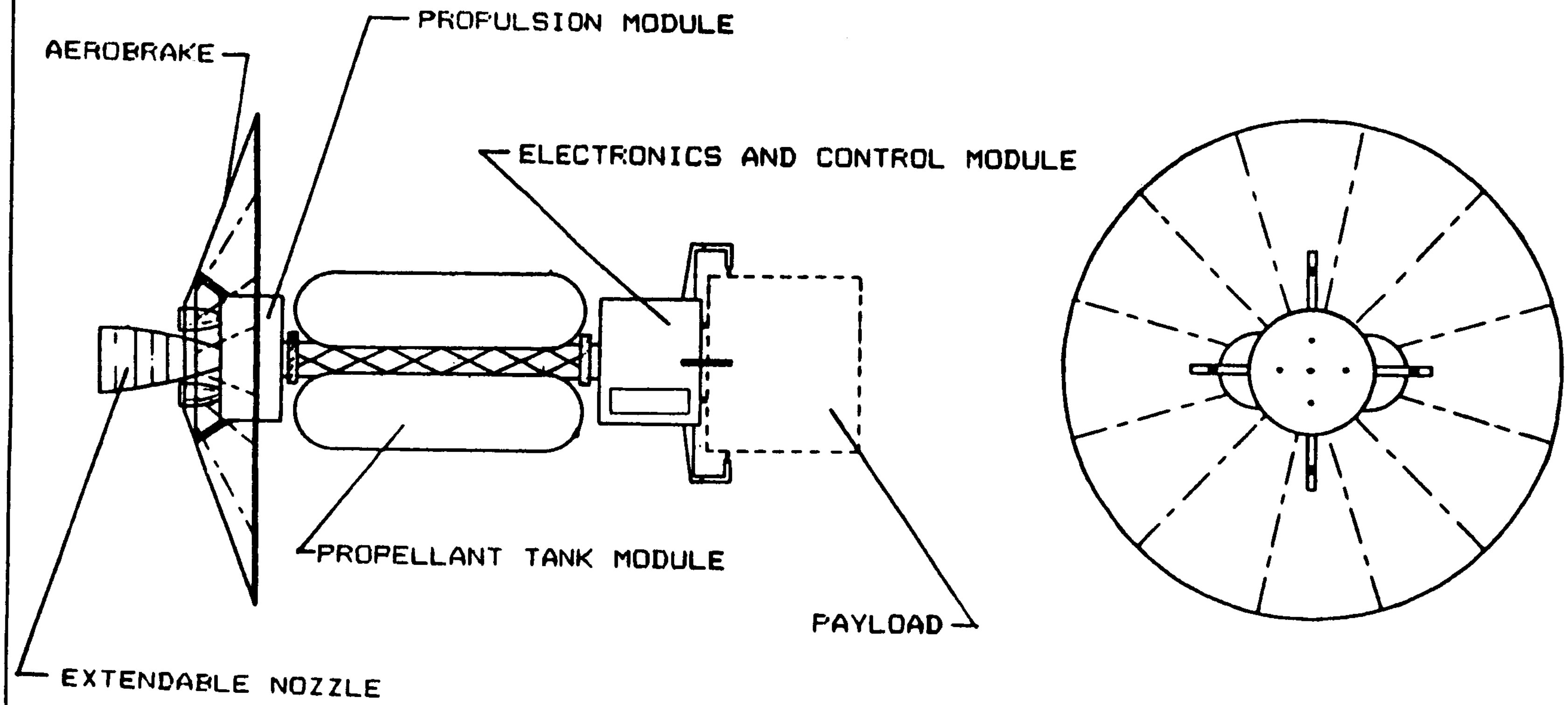



FIGURE 7. Original Proposal OTV Configuration

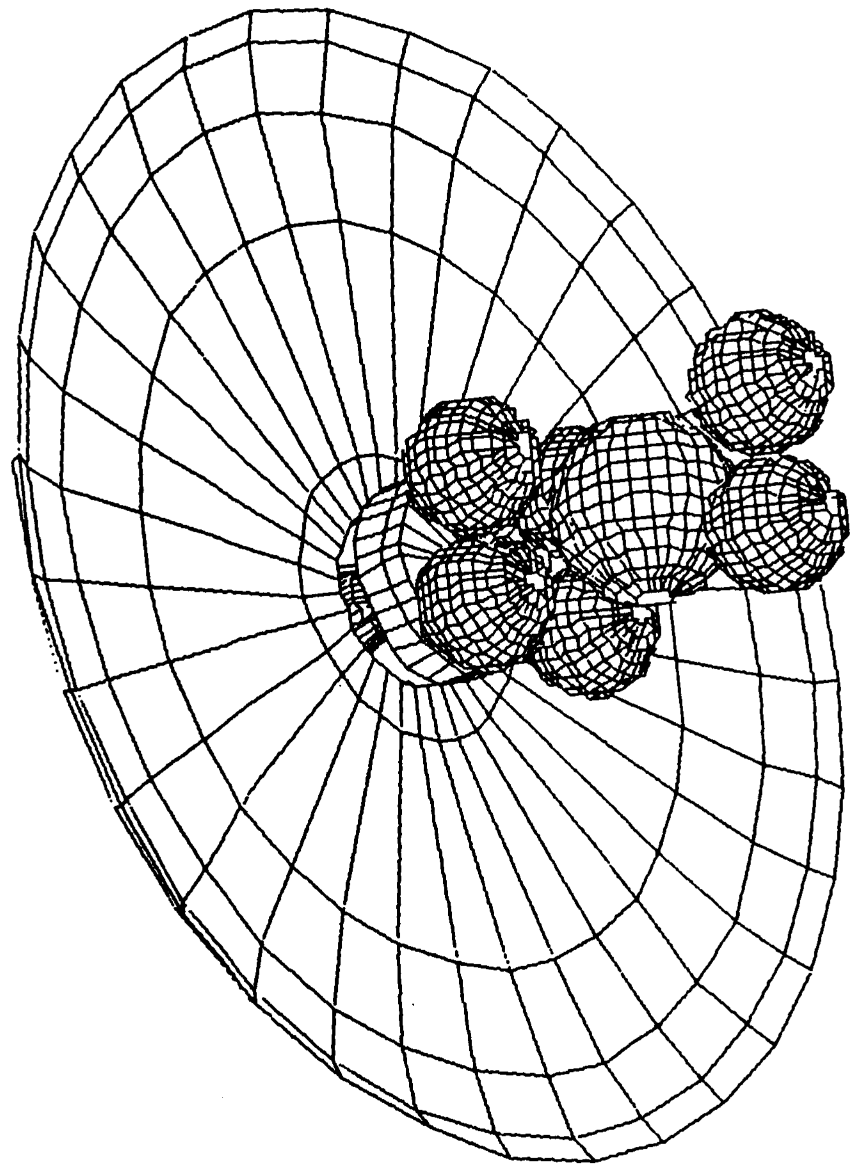


FIGURE 8. Multiple Hydrogen Tank OTV Configuration

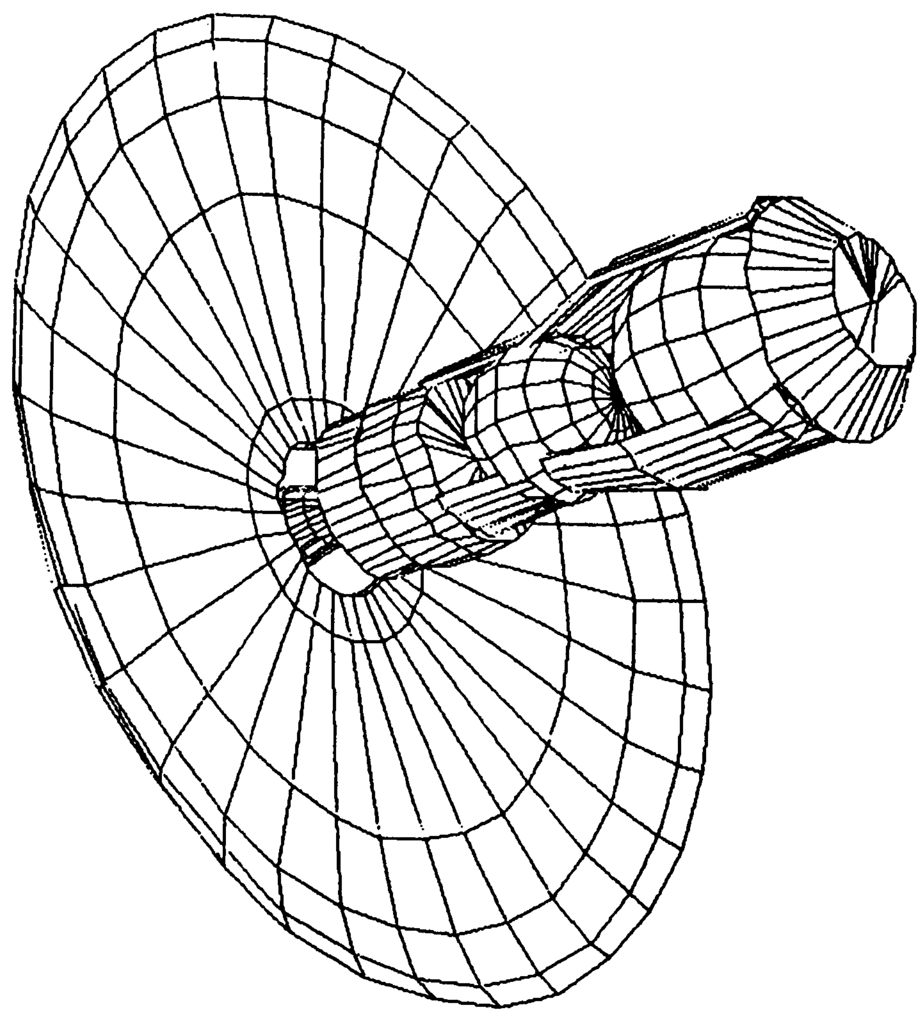


FIGURE 9. Double Spherical Tank OTV Configuration

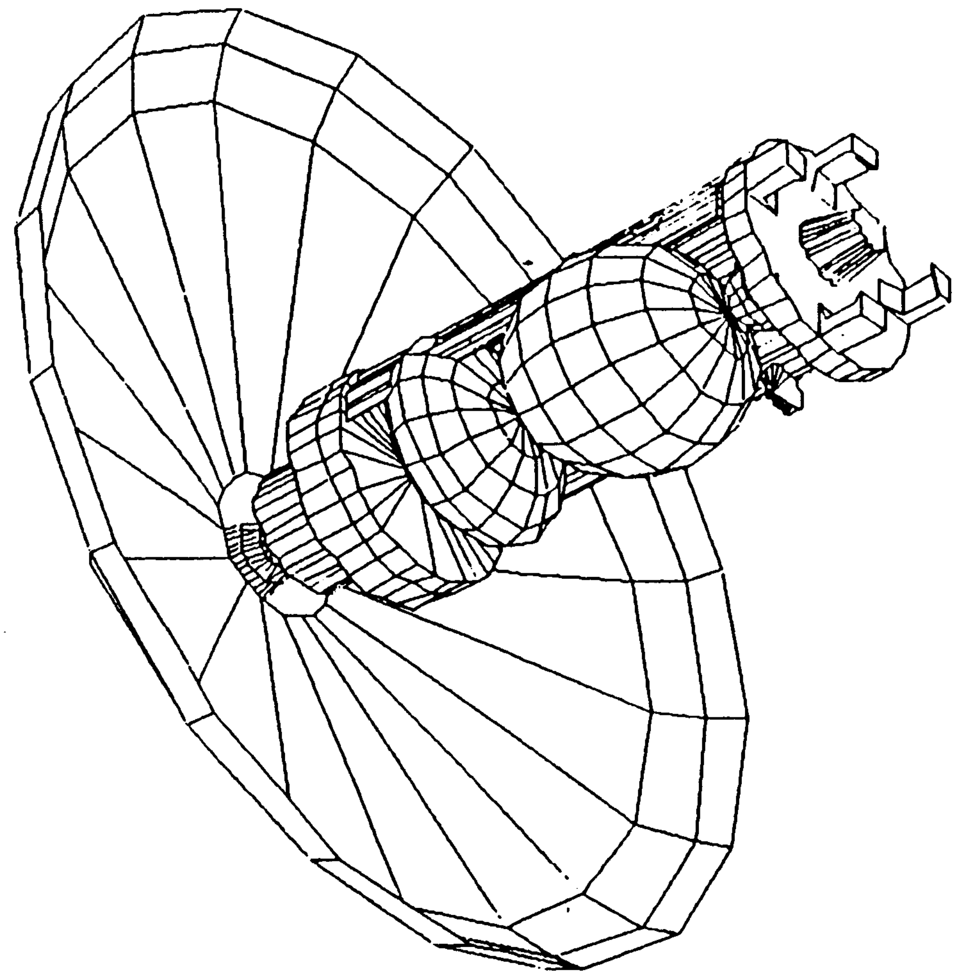


FIGURE 10. Three-D View of Final OTV Configuration

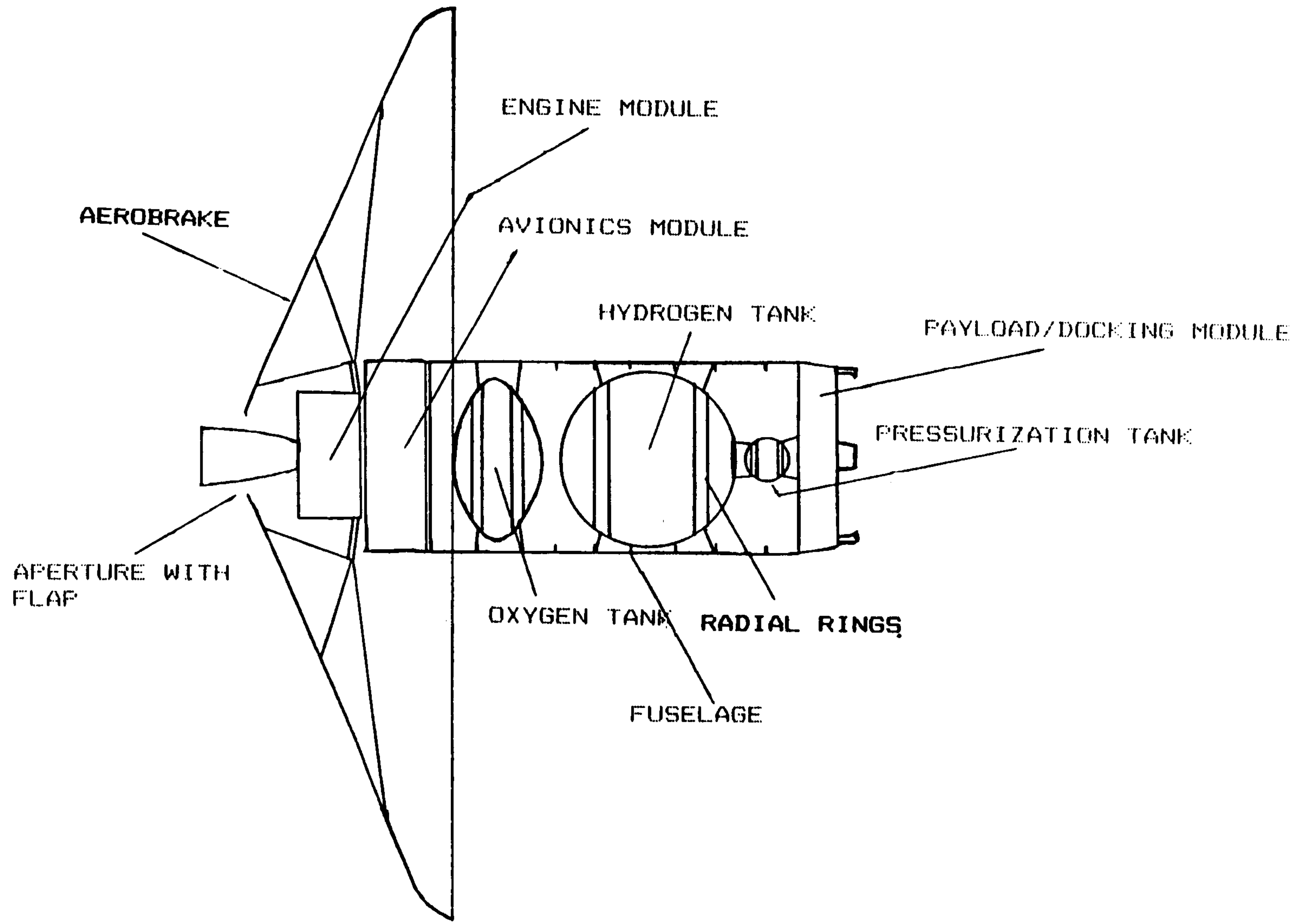



FIGURE 11 Cutaway Side View of OTV Structure

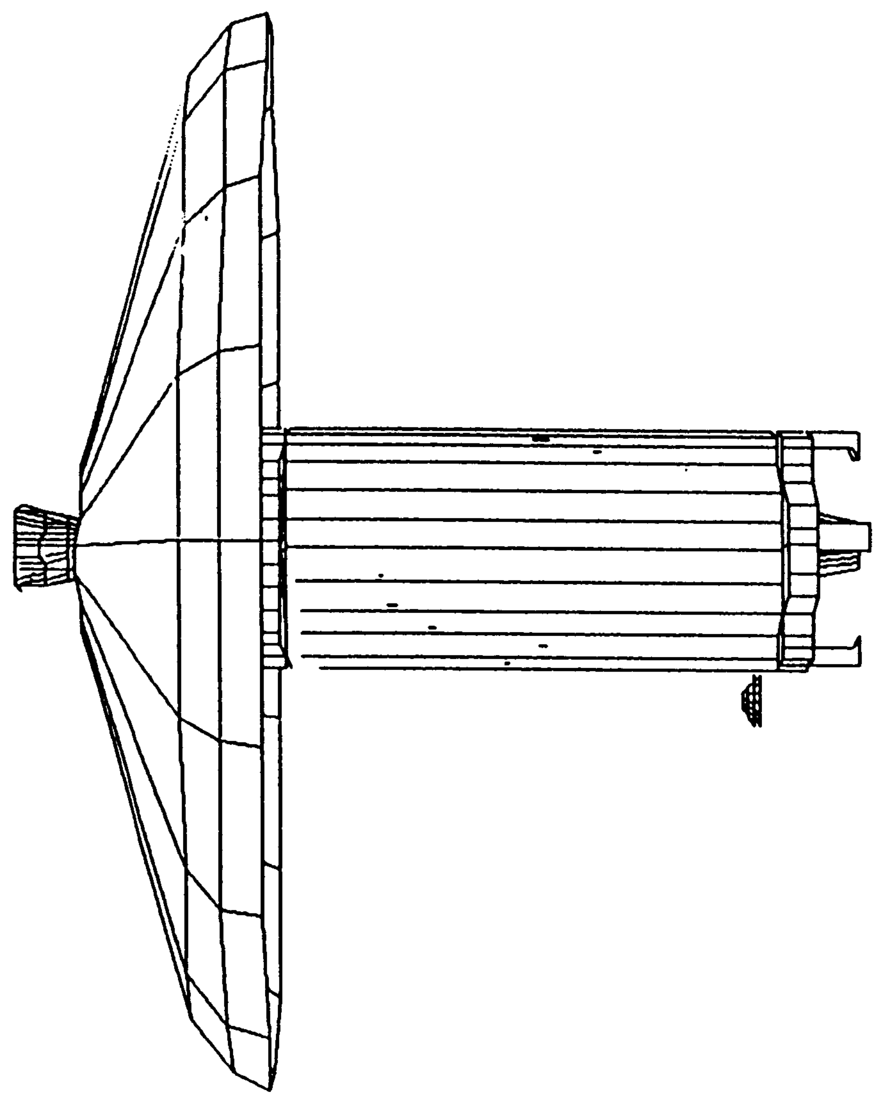


FIGURE 12 Side View of OTV Structure

| Payload Up (lbf) | Payload Down (lbf) | Total Fuel Required (lbf) All-Propulsive | Total Fuel Required (lbf) Aerobraked |
|---|---|---|---|
| 16000 | 0 | 65373 | 46692 |
| 0 | 18000 | 113109 | 47366 |
| 10000 | 7000 | 84287 | 47304 |
| 7000 | 10000 | 91718 | 46891 |
| 12000 | 5000 | 79332 | 47578 |
| 5000 | 12000 | 96673 | 46617 |

| Payload Up | Payload Down | Fuel Required for Each Burn LEO-ETO | ETO-GEO | GEO-ETO | ETO-LEO |
|---|---|---|---|---|---|
| 16000 | 0 | | | | |
| All-Prop: | | 34934 | 19193 | 6587 | 4658 |
| Aerobraked: | | 27561 | 15143 | 3988 | -- |
| 0 | 18000 | | | | |
| All-Prop: | | 47459 | 26075 | 23182 | 16393 |
| Aerobraked: | | 21513 | 11820 | 14033 | -- |

Table 1: Comparison of All-Propulsive versus Aerobraked Fuel Requirements for Possible Missions

```
                   1988  1989  1990  1991  1992  1993  1994  1995

Aerobraking R&D    *--------------------------*
Begins

Propulsion System
R&D Begins           *------------------*

Avionics Module
Development Begins       *------------*

Propellant Tanks
R&D Begins                      *------------*

Component R&D
Completed                                    **

First OTV Completed
and Prepared for
Space Trials                                        **

First OTV
Operational                                                   **
```

Table 2:  Time Line For Initial OTV Production